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Evaluating Advanced Propulsion Systems for the Titan Explorer Mission*†

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Early in 2001, several NASA centers and industry partners lead by the NASA Marshall Space Flight Center (MSFC) were asked to evaluate the performance of various advanced in-space propulsion systems. Over 30 different future missions were identified and an agency-wide study to assess each of these missions against a set of 5 to 20 different propulsion options was performed. Due to its priority and complexity, the Titan Explorer Mission is considered to be representative of the solar system exploration missions assessed. During 3 weeks, the Jet Propulsion Laboratory (JPL) Advanced Project Design Team ("Team X") performed a detailed analysis of this mission with the participation of technologists, mission analysts and system engineers involved in the agency-wide study. The advanced propulsion systems considered for Titan Explorer in this study included Aerocapture, Solar Electric Propulsion, Nuclear Electric Propulsion, Solar Sails, Mini-Magnetospheric Plasma Propulsion (M2P2) and Nuclear Thermal Propulsion. This paper summarizes the assumptions, technologies and findings for the Titan Explorer mission. Results are compared to an all state-of-the-art chemical propulsive case. This study shows that most advanced propulsion modules shorten the trip time of the mission by 2-3 years over an all-chemical approach and enable a reduction to a Delta IV Medium type launch vehicle. Promising technologies for this mission are Aerocapture, SEP and M2P2.

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Introduction

The work described in this paper is the result of a NASA agency-wide effort lead by Marshall Space Flight Center early 2001. The focus of this Integrated In-Space Transportation Planning (IISTP) study was to perform an evaluation of the performance and cost benefit of several advanced propulsion technologies applied to deep space missions. The ultimate objective was to help NASA's Office of Management and Budget (OMB) directs their propulsion augmentation funding. In order to examine the utility of each proposed option, a sample mission was considered which involved sending a science Orbiter and Lander to Saturn's moon, Titan. NASA/JPL led this task in cooperation with the other study partners, and used the JPL integrated design team called Team X. Team X had already worked in February 2001 on a Titan Orbiter and Lander study sponsored by Roy Kakuda [1]. This study became the basis for which the following propulsion options were evaluated:

- Chemical propulsion (pivot case), combined or not with Aerocapture,
- Solar Electric Propulsion (SEP), combined with Aerocapture,
- Nuclear Electric Propulsion (NEP),
- Solar Sail, combined with Aerocapture,
- Mini-Magnetospheric Plasma Propulsion (M2P2), combined with Aerocapture,
- Nuclear Thermal Propulsion (NTP).

The question to answer during this process was: "if that technology was available, what mission benefit would it have compared to the All-Chemical pivot case?" Thus the design team accepted the technology as presented by the technologists and no efforts were made here to assess the feasibility of the development of the technology in the timeframe considered. The main objective for each of the propulsion option was then to minimize trip time and launch vehicle cost.

This paper first describes the Titan Explorer reference mission and the variations in the orbiter's system design that were implied by the use of the various propulsion systems. The advanced propulsion technologies considered and related mission design and assumptions are then described. Finally, a summary of overall system performances and conclusions are presented.

Titan Explorer Mission and System Design

The Titan Explorer reference mission is comprised of an Orbiter and a Lander. The objectives of the orbiter mission are to provide a global mapping of Titan surface. The Orbiter will carry as science instruments a SAR/altimeter, a radio science instrument (USO), a narrow angle, and an infrared radiometer. The objectives of the Titan Lander will be to provide data on the distribution and composition of the surface organics, the organic chemical processes, their chemical context and energy sources, prebiological or protobiological chemistry, geological and geophysical processes and evolution, atmospheric dynamics and meteorology, and seasonal variations and interactions of the atmosphere and surface. The Lander strawman payload includes a GCMS, UV/visual/near IR line spectrometer, near angle and wide angle imagers, 2 chemistry labs, XRFS, entry ASI, descent altimeter, sample acquisition and handling system.

The reference mission includes a 3-year science mission after circularization around Titan. Titan's final orbit is a 1400-km altitude circular orbit. The launch date is tentatively proposed for 2010. Based on the long mission life and desired reliability, a full-redundancy, Class A mission is assumed.

Throughout the study, small design changes were made to the reference mission Orbiter design to accommodate the requirements driven by the propulsion system. The reference mission Lander design was never affected by the propulsion system therefore is was treated as a black box of fixed mass.

A variety of mission trajectories were considered based on the capabilities of the propulsion system and the most efficient use of the launch vehicle resources. For most cases, the propulsion system was designed as a separate stage from the Orbiter/Lander and was used primarily for the trans-Saturn injection. In these cases, the stage was separated during cruise to Saturn and the Orbiter/Lander were inserted into the 1400 km circular orbit around Titan using aerocapture and orbit circularization with an Hydrazine propulsion system. In two cases (NEP and NTP-bimodal), this approach was not used because the Orbiter and primary propulsion system were designed as an integrated unit. The justification for this approach was that these systems also provided electrical power to the Orbiter.

Table 1 summarizes the "payload" (with respect to the propulsion system) for each of the three system architectures considered to reach and circularize around Titan. The "all propulsive chemical" case is the case where only chemical propulsion is used to reach Titan's final orbit. The "all propulsive nuclear" includes to sub-cases: the Nuclear Electric and Nuclear Thermal Bimodal propulsion systems. All the other propulsion options used aerocapture to circularize around Titan.

The rationales for the variations in Orbiter and propellant masses are also summarized in Table 1. In the NEP or NTP Bimodal options, it was assumed that the nuclear power system would also provide power to the spacecraft. Therefore the four Radioisotope Power Sources (RPS) were removed from the Orbiter's reference design. However, a scan platform and additional gimbals were deemed necessary to provide the pointing accuracy required by the science instruments and the telecommunication antenna. In the case of aerocapture, a ΔV of 600 m/s was assumed for post-aerocapture circularization and operations at Titan. The initial orbit after the aerocapture maneuvers was assumed to be 800 km by 42,000 km altitude. This high elliptical orbit reduced the atmospheric ΔV requirements on the Aerocapture system. Thus the 600 m/s include 13 m/s for periapsis raise to 1400 km, 534 m/s for apoapsis decrease to 1400 km, 27 m/s for gravity losses (5%) and 26 m/s of contingency.

For all the propulsive options, 30% mass and 30% power contingencies were applied to all spacecraft subsystems, and a 10% launch vehicle margin was assumed (consistent with JPL conceptual design guidelines). However, only 5% power contingency was applied to the SEP and nuclear power subsystems, after degradation. The structures/cabling masses are not based on a specific design but are a percentage of the subsystems to which the structures apply (typically 26% of the propulsion system and 16% of the power system for structures). These percentages are based on historical data and are consistent with the design guidelines of the JPL integrated project design center (Team X). The JPL IISTP Team X report [2] provides additional details on the propulsion, attitude control, power, thermal, structure and mission operations sub-systems for each of the propulsion options considered.

Table 1: Payload mass in kg to insert into Titan's final orbit as a function of system architecture.

Mass in kg	**All Propulsive Chemical**	**All Propulsive Nuclear**	**Aerocapture**
Lander mass:	235	235	235
Contingency (30%):	70	70	70
Orbiter mass:	405	380	420
Contingency (30%):	120	115	125
Propellant/pressurant:	20	50	250
Total:	**850**	**850**	1110
Aerocapture System:	N/A	N/A	370*
Total Payload Mass:	**850**	**850**	**1380**
Orbiter system comments:	- 4 RPS Stirling	- Scan platform for science instruments - Telecom Antenna gimbals	- 600 m/s post-aero-capture ΔV - 4 RPS Stirling

* For an aerocapture mass fraction of 25%, dependent on the entry velocity.

Propulsion Systems and Mission Description

All propulsion technologies considered here are based on predictions by the various technologists involved of the state of the technology by 2010, assuming funding would be appropriately assigned/allocated. No effort was made to verify the validity of the predictions. Also, the level of funding to reach the Technology Readiness Level 6 (TRL 6 is...) by 2010 is significantly different between technologies, but that was not a determining factor in this mission evaluation of the benefits of the technologies.

1. State-of-the-art Chemical Propulsion

Technology

The state-of-the-art chemical propulsion system evaluated here features hydrazine-N_2O_4 engines (LEROS 1-C type), which have an Isp of 325 s, composite over-wrapped (COPV) propellant tanks, and composite helium pressurant tanks. This pressurized bipropellant propulsion system was chosen to maximize performance and minimize initial mass. The number of engines and size of the tanks were dependent on the chemical option studied.

Trajectory

The Earth-Saturn trajectory was developed by Ted Sweetser, Team X leader. As shown on Figure 1, this trajectory has three Venus Gravity Assists and an 8.4-year flight time. Launch is in July 2010 and arrival in December 2018. The launch C3 is 10.9 km^2/s^2. The arrival Vinf at Saturn is 9.0 km/s and at Titan 7.3 km/s. The entry velocity (inertial) at Titan is 7.7 km/s at 800 km altitude. The post-launch ΔV to approach Saturn/Titan was estimated at 1.85 km/s including margins. An additional 5.98 km/s was needed for a chemical insertion to titan 1400-km altitude final orbit. However, the work done for the Europa Orbiter [3] indicates that a moons tour design around Saturn could probably reduce the ΔV to about 3 km/s for insertion around Titan. This would increase the flight time by about 2 years.

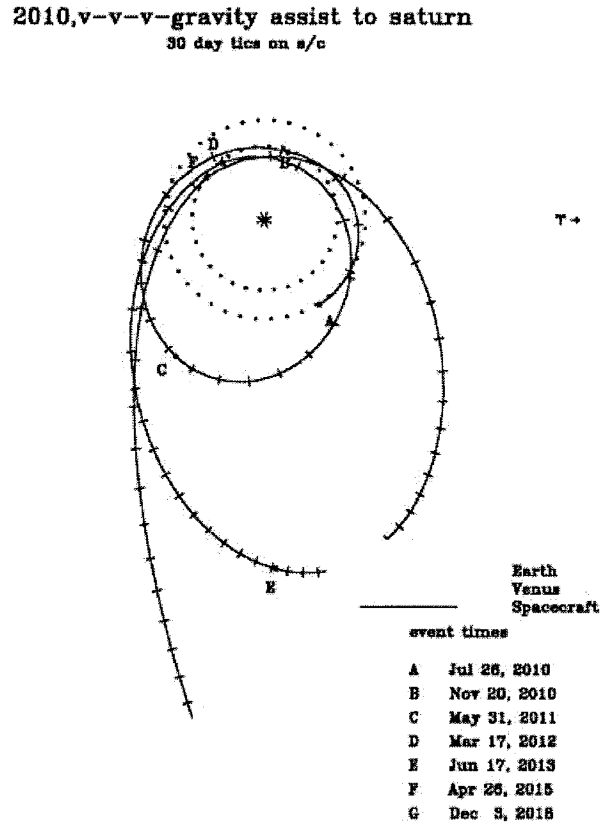

Figure 1: Triple Venus Gravity Assist to Saturn trajectory

System

Two mission architectures were evaluated for this technology: the first option is an all-chemical insertion to Titan. Since the total ΔV for this option was considerable (1.85 + 3 km/s), the chemical propulsion module was staged. The first stage performed the first 1.85 km/s, and second stage the 3 km/s. Table 2 provides a mass breakdown of this option. The smallest launch vehicle this spacecraft requires a Titan IV Centaur or bigger.

The second option uses chemical propulsion to reach Saturn and aerocapture to insert into Titan. The aerocapture system mass fraction was provided by the aerocapture technologists and in this case was 25% of the total entry mass. Table 3 summarizes the spacecraft mass breakdown. This option uses an Atlas 531, which delivers 3850 kg (with 10% margin), at a C3 of 10.9 km^2/s^2.

Table 2: All Chemical Propulsion spacecraft mass breakdown

Subsystem	Mass (kg)	Comment
Payload	850	
Attitude Control	5	
Power	0	
Thrusters	8	2 engines
Tanks, Feed System	241	
Total Structure	605	
Cabling	23	
Thermal	65	
Total:	**947**	
Contingency (30%)	284	Not incl. aero
Propellant	6126	Incl. residuals
Adapter	65	
Launch Total:	**8272**	

Table 3: Chemical Propulsion / Aerocapture spacecraft mass breakdown

Subsystem	Mass (kg)	Comment
Payload	1110	
Attitude Control	5	
Power	0	
Thrusters	8	2 engines
Tanks	61	
Feed System	5	
Total Structure	232	
Cabling	20	
Thermal	25	
Aeroshell System	370	
Total:	**726**	
Contingency (30%)	107	Not incl. aero
Propellant	1667	Incl. 5% resid.
Adapter	65	
Launch Total:	**3675**	LV = 3850 kg

2. Solar Electric Propulsion

Technology

Two advanced ion propulsion technologies were considered: a 30-cm ion thruster capable of processing 5 kW at a specific impulse (Isp) of 5000 seconds, and a 40-cm ion thruster processing 10 kW at 3800 s. Their comparison would help guide future efforts in that field. Both engines use Xenon for propellant. For state-of-the-art ion propulsion technology, please refer to [3]. Table 4 summarizes each ion propulsion technology assumptions. Both options used aerocapture for insertion around Titan.

Table 4: Comparison of the 5-kW and 10-kW thruster and system technologies

	5-kW thruster	10-kW thruster
Power range (kW)	1-5	1-10
Engine diameter (cm)	30	40
Isp (sec)	2000-5000	2500-3800
Xe throughput (kg)	200	500
Mass (kg)	7	12
PPU mass (kg)	15	27
Heritage	NSTAR	NSTAR

The thrusters are powered by the Power Processing Units (PPUs), which convert the power from the solar arrays to the voltages and currents required by the engine. The feed system and PPUs are controlled by a Digital Control Interface Unit (DCIU), which accepts and executes high-level commands from the spacecraft computer and provides propulsion subsystem telemetry to the spacecraft data system. New PPU and DCIU designs were assumed. The solar array technology considered was the AEC-Able Ultraflex design that would carry 29% efficient Triple-Junction cells. The specific power of this array is 178 W/kg at beginning-of-life (BOL).

Trajectory

The low-thrust Earth-Saturn trajectories were calculated by Carl Sauer at JPL. With the appropriate thruster models, trajectories were run parametrically as a function of trip time for a fixed SEP optimum power of 24 kW (end-of-life (EOL) equivalent at 1 AU). All trajectories included a Venus Gravity Assist. Figure 2 shows an example of such a trajectory. For both thrusters, the chosen trajectory was the one that minimized trip time and fits within the launch vehicle capabilities, the Delta 4240.

For the 5-kW thruster case, the launch C3 is 10.3 km^2/s^2. The arrival Vinf at Saturn is 9.3 km/s and at Titan 7.5 km/s. The entry velocity (inertial) at Titan is 7.9 km/s at 800 km altitude. The low thrust ΔV is 8.6 km/s. The trip time is 5.2 years.

For the 10-kW thruster case, the launch C3 is 5.8 km^2/s^2. The arrival Vinf at Saturn is 9.1 km/s and at Titan 7.4 km/s. The entry velocity (inertial) at Titan is 7.8 km/s at 800 km altitude. The low thrust ΔV is 9.6 km/s. The trip time is 5.5 years.

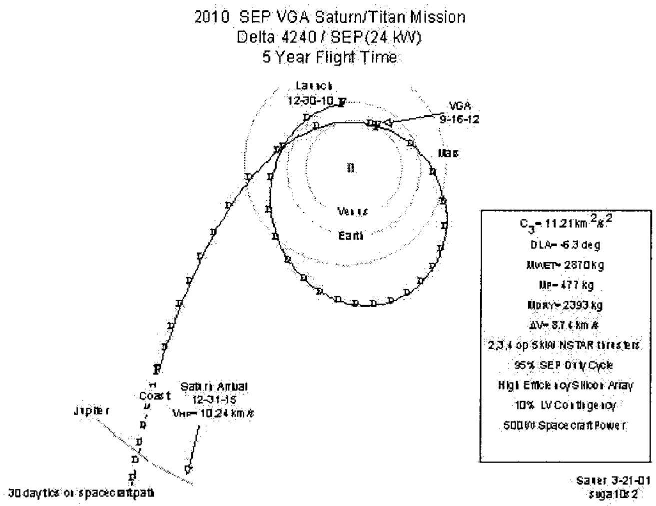

Figure 2: SEP Venus Gravity Assist to Saturn trajectory, 5-year trip time.

System

For both cases the SEP module was jettisoned prior to aerocapture. The aerocapture system mass fraction was 25% of the total entry mass.

For the 5-kW thruster case, 4 operating thrusters and PPUs were needed. The 10-kW thruster case used 2 operating thrusters and 2 operating PPUs. One spare ion engine, one spare PPU and DCIU were also included for single-fault tolerance. Each thruster was gimbaled separately.

The 24-kW EOL at 1 AU solar array design has two wings of 7.2 m diameter. Also, in order to support power demand during launch, a primary battery was used prior to solar array deployment.

The engines control the spacecraft in pitch, yaw and roll during SEP operations. When the SEP system is not operating but is attached, the hydrazine monopropellant system on the orbiter stage performs attitude control.

The tankage fraction was calculated assuming cylindrical composite tanks. Those tanks have a propellant storage efficiency (Tank Fraction TF) of about 2.5% for Xenon when stored as a supercritical gas (~2000 psia). Furthermore, a 10% propellant contingency was added to the deterministic propellant mass to account for residuals, attitude control and margin. The launch vehicle adapter was assumed to be integrated with the SEP module structure. Table 5 and 6 summarize the total spacecraft mass breakdown.

Table 5: SEP 5-kW Propulsion Module and Spacecraft Mass Breakdown

Subsystem	Mass (kg)	Comment
Payload	1110	
Attitude Control	4	
Power	171	
Thruster,PPU,DCIU	115	4 operating
Tanks	13	
Feed System	11	
Total Structure	268	
Cabling	46	
Thermal	68	
Aeroshell System	370	
Total:	**1066**	
Contingency (30%)	209	Not incl. aero
Propellant	527	10% conting.
Launch Total:	**2912**	LV = 2929 kg

Table 6: SEP 10-kW Propulsion Module and Spacecraft (payload) Mass Breakdown

Subsystem	Mass (kg)	Comment
Payload	1110	
Attitude Control	3	
Power	171	
Thruster,PPU,DCIU	122	2 operating
Tanks	21	
Feed System	7	
Total Structure	291	
Cabling	46	
Thermal	79	
Aeroshell System	370	
Total:	**1110**	
Contingency (30%)	222	Not incl. aero
Propellant	804	10% conting.
Launch Total:	**3246**	LV = 3239 kg

The Delta 4240 has a launch capability of 2929 kg at C3 = 10.3 km^2/s^2 and 3239 kg at C3 = 5.8 km^2/s^2.

3. Solar Sail

Technology

The Solar Sail technology considered here was based on the best current predictions of the performances that would be available in the 2010 time frame if sufficient funding was allocated [5]. Sails are large ultra-light mirrors, which use light from the Sun for low-thrust propulsion. The sail configuration chosen here is composed of four deployable sections (see Figure 3), with an areal density of 10 gm/m^2 (includes contingency). The deployed sail was 345 meters on a side, for a total area of around 118,000 m^2. This multi-sail configuration was proposed to achieve the necessary overall size within the constraints of the German designed DLR-style boom [6]. Each deployable section is composed of 4 triangular panels made of a 2.5 micron aluminized polyamide film. The sail film is compactly stowed for launch and deployed and supported by 4 deployable booms. The spacecraft module is attached to the sail with a 4-m articulated mast.

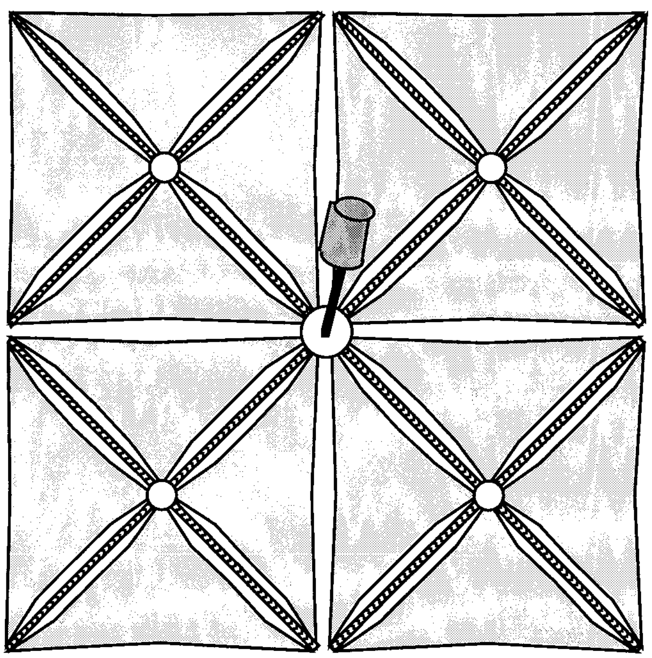

Figure 3: Solar Sail Configuration.

Trajectory

The Earth-Saturn sail trajectory was originated by Carl Sauer at JPL. It features 3.5 revolutions around the Sun with a minimum distance from the Sun of 0.46 AU. The flight time is 8.5 years. The launch energy is $C3 = 0\ km^2/s^2$. The characteristic acceleration is 0.37 mm/s2, and the arrival Vinf at Saturn is 9 km/s and at Titan 7.3 km/s. The entry velocity (inertial) at Titan is 7.7 km/s at 800 km altitude. Here again, aerocapture was used to perform Titan's orbit insertion. The launch vehicle is the Delta 4240 with a launch capability of 3667 kg at that C3.

System

The sail module was jettisoned prior to aerocapture. This module only carried the solar sail package and about 300 kg of deployment hardware that was jettisoned right after deployment. This mass was not taken into account in the trajectory calculations. The post-deployment solar sail module dry mass was about 1560 kg. The aerocapture system mass fraction was 25% of the total entry mass.

The Orbiter supplied all subsystem functionality for the sail. Also, since the trajectory goes close to the Sun, additional thermal control for the Orbiter was taken into account. And since the sail module carried the Orbiter/Lander mass during launch, additional mass compared to the original design proposed by the technologists was book-kept in the sail structure.

The spacecraft while sailing is steered by adjusting the center of mass with respect to the center of pressure (Cg/Cp shift). The spacecraft is placed on a boom, allowing it to be moved relative to the sail's center of pressure, which produces a torque used to change the sail's attitude. Slew maneuvers thus could be done in timeframes of hours to days. Table 7 summarizes the total spacecraft mass breakdown.

Table 7: Solar Sail Propulsion Module and Spacecraft (payload) Mass Breakdown

Subsystem	Mass (kg)	Comment
Payload	1138	
Attitude Control	17	
Avionics	18	
Cabling	8	
Mechanisms/deplmt	468	
Total Structure+film	579	
Thermal	16	
Aeroshell System	380	
Total:	**1486**	
Contingency (30%)	332	Not incl. aero
Propellant	20	
Adapter	65	
Launch Total:	**3041**	LV = 3667 kg

4. Mini-Magnetospheric Plasma Propulsion

Technology
The M2P2 technology considered here was based on the work currently done at the University of Washington [7]. This propulsion system creates a magnetic bubble around and attached to the spacecraft, which is then carried by the solar wind. A low energy plasma is used to inflate the magnetic field to a large cross section (15-30 km). As the spacecraft moves away from the Sun, the magnetic bubble expand such that the force exerted on the spacecraft is constant.

The plasma is created by a RF antenna located inside a permanent magnet. A feed system provides Xenon (other propellants are possible, but Xe was used here for simplicity of storage) inside the magnet (see Figure 4). The antenna and plasma frequency is 13 MHz, which should not disrupt the telecommunications subsystem. As for SEP, solar arrays, AEC-Able Ultraflex type, provides power to an 85% efficient RF system, which then route about 2 kW of AC power to each thruster.

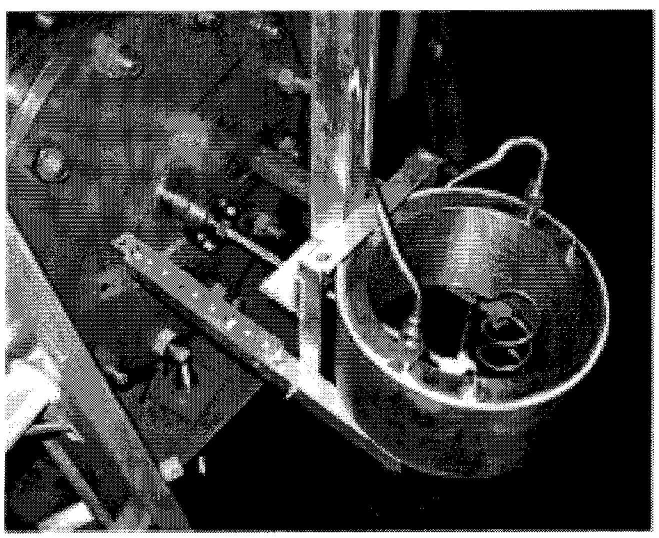

Figure 4: M2P2 Prototype.

Trajectory
The Earth-Saturn M2P2 trajectory was computed by Ted Sweetser, Team X leader. It is a direct trajectory to Titan, starting from Earth escape (C3 = 0 km^2/s^2). The flight time is 5.6 years. A tilt of 5^o in the thrust was assumed in order to thrust out of the ecliptic and therefore avoiding a 426 m/s broken plane maneuver. The thrusting lasts for 332 days up to 3.2 AU. It used Xenon at the rate of 0.5 kg/day/kWe.

The arrays were sized to provide 1.2 times the power needed at 1 AU, after which the thrusters are throttled down as 1/r2. The total power needed from the array was about 9.2 kW (7.2 kW used for the trajectory calculations). At about 2.25 AU, the thrusters are at their lower operating point (400 W). Thus after 2.25 AU, the thrusters were pulsed, sending "puffs" of plasma out into the mini-magnetosphere.

The arrival Vinf at Saturn is 5.9 km/s and at Titan 4.7 km/s. The entry velocity (inertial) at Titan is 5.35 km/s at 800 km altitude. Aerocapture was also used to perform Titan's orbit insertion. The aerocapture system mass fraction was 20% of the total entry mass, since the entry velocity was lower than the other entry cases.

The launch vehicle is the Delta 4240 with a launch capability of 3667 kg at C3 = 0 km^2/s^2.

System
The M2P2 system considered for this mission is composed of 3 redundant thrusters. The redundancy is achieved at a component level with 2 antennas per thruster and a mechanism that rotates the second antenna by 180^o to switch the active antenna and align its waveguide feed with the supply waveguide. The 3 pairs of M2P2 propulsion thrusters are attached to the Orbiter through 3 separate 3-m long booms. Each thruster pair is mounted to a 2-axis actuator. Figure 5 depicts the spacecraft configuration.

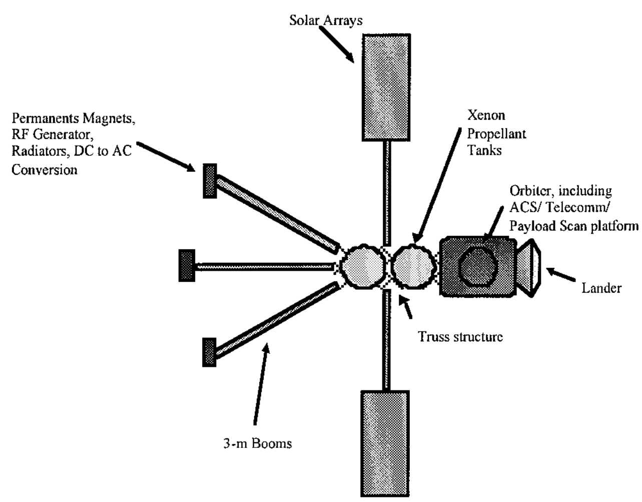

Figure 5: M2P2 Spacecraft Configuration.

In addition, the three booms host at their ends the RF PPU and a large radiator (~ 1.3 m^2) and heat pipe to cool the RF PPU and the magnets.

The outward ends of the 3 booms lie at the points of an equilateral triangle that is several meters on a side. The triangle can be thought of as the base of a tetrahedron formed by the booms. Each M2P2 unit interacts with the solar wind and by adjusting the direction of the three individual thrust vectors, the M2P2 units are used to provide control torque in all 3 axes until they are jettisoned prior to arrival at Titan. The thrusters are capable of causing a resultant force inclined 5 to 10 degrees with respect to the Sun-probe radius. They provide about 3 N of resultant force on the spacecraft.

Each thruster was 15 kg, antenna 0.5 kg, the RF PPU was estimated at 30 kg per thruster, and DCIUs at 5 kg each (2 DCIUs, 1 redundant). Table 8 shows a summary of the spacecraft mass breakdown.

Table 8: M2P2 Propulsion Module and Spacecraft Mass Breakdown

Subsystem	Mass (kg)	Comment
Payload	1110	
Attitude Control	7	
Power	67	
Thruster,RF PPU...	143	3 operating
Tanks	11	TF = 2.5%
Feed System	12	
Total Structure	259	
Cabling	41	
Thermal	86	
Aeroshell System	280	
Total:	**906**	
Contingency (30%)	189	Not incl. aero
Propellant	462	10% conting.
Adapter	65	
Launch Total:	**2732**	LV = 3667 kg

5. Nuclear Electric Propulsion

Technology

A Nuclear Electric System is a complex system composed of several subsystems. A block diagram of the NEP systems used in this study is provided in Figure 6. The overall NEP vehicle configuration is based on the use of a nuclear reactor that generates thermal power, which is converted into electric power to feed an ion propulsion system. In general, a long boom separates the power and propulsion systems from the other subsystems of the spacecraft to reduce the radiation dose created by the reactor. The boom also serves as a structural attachment for the deployable radiators (flat or conical). Every elements of the vehicle other than the reactor are located in the reactor shield's shadow. The power conversion system, propulsion system fuel tanks, feed system, power processing and thrusters are mounted next to the shield. The very large deployed radiators are unfolded along each side of the main boom or in other design form a cone around the structural boom and follow the shield shadow. More details on NEP configuration can be found in [8] and [9].

Power System

This study considered 2 types of nuclear power systems. The propulsion system required a power of 90-kWe into the thrusters. One power System was based on a 100-kWe liquid metal cooled nuclear system, and the other one on a 100-kWe Heat-pipe cooled system. Both system characteristics are summarized in Table 9. Both systems should set reasonable boundaries on the expectation of the power system specific masses for use in the 2015 timeframe.

Table 9: Comparison of the Liquid metal cooled and Heat Pipe Cooled NEP system technologies

	Liquid Metal	**Heat Pipe SAFE-300**
Power (kWth)	450	350
Electric power (kWe)	100	100
Nuclear Fuel	U Nitride	UO_2
Turb. inlet temp. (K)	1300	1200
Conversion cycle	Brayton	Brayton
Conversion Efficiency	22%	28%
Thermal radiator tech	Carbon HP	CPL/LHP
Radiator specific mass	6 kg/m^2	5 kg/m^2
Radiating area (m^2)	95	90, 2 sided
Radiator temperature	600 K	480 K
Shield	10^5 krads @ 2 m	200 krads @ 12 m
Power system mass	2641 kg	1993 kg
Power syst. spec. mass (not incl. structure)	26 kg/kWe	20 kg/kWe

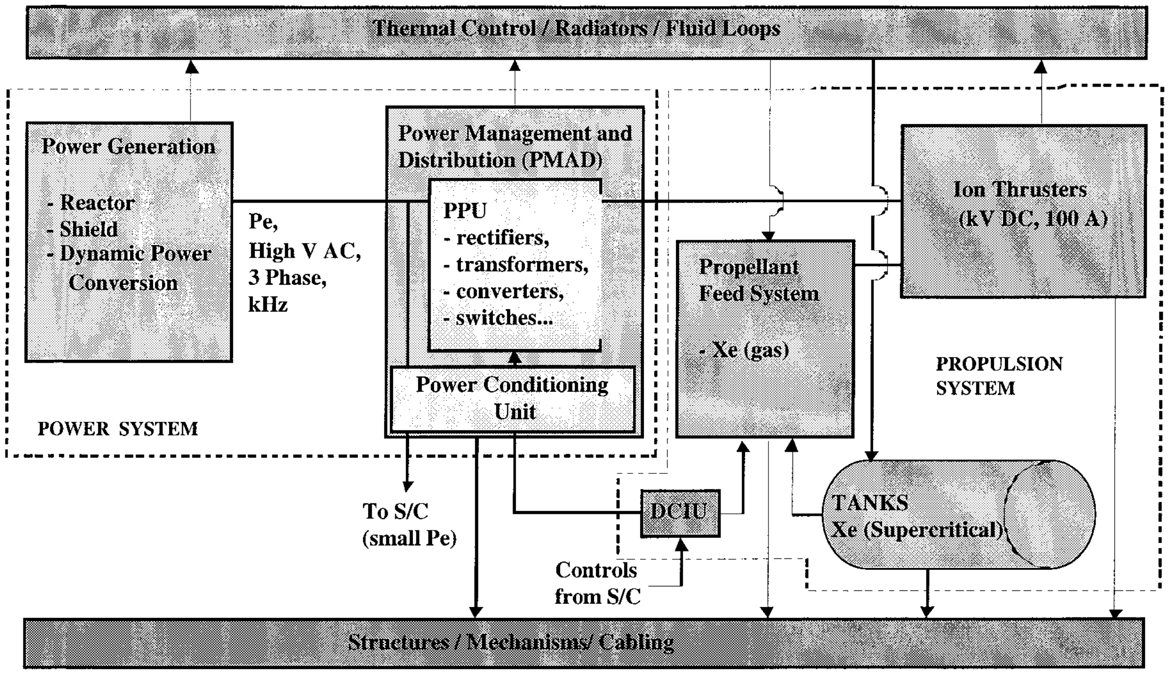

Figure 6: NEP System Block Diagram.

Ion Propulsion System

The ion propulsion system (IPS) is composed of 50-cm diameter ion engines that can process 30-kW of electric power and use Xenon as propellant. The thruster has an assumed efficiency of 0.75 at a specific impulse (Isp) of 9000 s. The propellant throughput capability of each engine was assumed to be 750 kg. The thrusters are powered by the Power Processing Units (PPUs), which convert the power from the turbo-alternator to the voltages and currents required by the engine. The mass and complexity of the PPUs were greatly reduced by tuning the output voltage of the turbo-alternator to a value close to the thruster input demand (direct-drive architecture). The design of the turbo-alternator was assumed flexible enough to allow for this tuning. The efficiency of the PPUs was estimated at 0.94. Each PPU processes 33 kW of power.

The feed system and PPUs are controlled by the Digital Control Interface Unit (DCIU), which accepts and executes high-level commands from the spacecraft computer and provides propulsion subsystem telemetry to the spacecraft data system.

All elements of this propulsion system would be new design with some technology heritage from the NSTAR ion propulsion system that flew on Deep-Space 1. Each thruster was estimated at 20.9 kg, Direct-Drive PPU at 30 kg per thruster, feed system at 10 kg per thruster, and DCIUs at 2.5 kg each.

For both nuclear systems considered, 3 operating thrusters and PPUs were needed for power requirements, but 4 thrusters were used for throughput requirements. And as for the SEP cases, one spare ion engine, one spare PPU and DCIU were also included for single-fault tolerance. Each thruster was gimbaled separately.

Trajectories

The NEP trajectories were run by Leon Gefert at NASA GRC. They were run parametrically as a function of flight time using the characteristics of the ion thrusters described above. The starting point was a Low Earth Orbit, circular at 2500-km altitude. This altitude was chosen to be compliant with the NASA Orbital Debris Guidelines in case the system failed to start. The NEP vehicle spirals out of the Earth in about 1 year, goes directly to Titan and spirals down around Titan to its final orbit (80-100 days). The total trip time for the Liquid Metal Cooled power system was 7.6 years with a total low thrust ΔV of 34.0 km/s, and 6.7 years for the Heat-pipe Cooled power system with a low thrust ΔV of 34.7 km/s.

The initial mass for the Liquid Metal Cooled power system was 8488 kg (with 10% derating), and 7208 kg for the Heat-pipe Cooled power system. The launch vehicle that provides sufficient injected mass for these two options is the Delta 4450.

Systems
The NEP module included necessary attitude control, structure, and thermal subsystems as well as body mounted solar arrays and batteries to provide power before reactor start. The Orbiter supplied command and data handling, telecommunications and some additional attitude control.

The dimensions of the NEP vehicle brought challenges for attitude control and pointing of the telecommunication subsystem, which lead to the addition of a boom and gimbals for the antenna. A scan platform was also required for pointing of the science instruments.

Tables 10 and 11 show a summary of the spacecraft mass breakdown.

Table 10: NEP Propulsion Module and Spacecraft Mass Breakdown – Generic Liquid Metal Cooled Power System

Subsystem	Mass (kg)	Comment
Payload	850	
Attitude Control	13	
Power System	2435	
Reactor & Shield	747	
Heat exchanger	155	
Power conversion	823	
Heat rejection	547	
Power managment	163	
Thruster,PPU,DCIU	229	5 thrusters
Tanks	75	TF = 2.5%
Feed System	50	
Total Structure	606	
Cabling	95	
Thermal	52	
Total:	**3555**	
Contingency (30%)	1067	
Propellant	3016	10% conting.
Launch Total:	**8488**	LV = 8585 kg

Table 11: NEP Propulsion Module and Spacecraft Mass Breakdown – SAFE-300 Power System

Subsystem	Mass (kg)	Comment
Payload	850	
Attitude Control	13	
Power System	1826	
Reactor & Shield	670	
Heat exchanger	50	
Power conversion	560	
Heat rejection	456	
Power managemt	90	
Thruster,PPU,DCIU	229	5 thrusters
Tanks	65	TF = 2.5%
Feed System	50	
Total Structure	554	
Cabling	95	
Thermal	52	
Total:	**2884**	
Contingency (30%)	865	
Propellant	2609	10% conting.
Launch Total:	**7208**	LV = 7295 kg

6. Nuclear Thermal Propulsion

Technology
The NTP system consists of a nuclear reactor, a large liquid Hydrogen tank and an exhaust system composed of 37 individual nozzles through which the Hydrogen is expanded (see Figure 7). The thermal energy from the fission reactor is applied to the propellant. The state-of-the-art engine has demonstrated a specific impulse of 845 seconds. The projections for this system that were taken into account in this study were an Isp of 940 s. The Nuclear Thermal Rocket (NTR) considered had a 75 MWth reactor and provided 27 kN of thrust.

This system could be used either on a regular mode, where the engine would burn only for a few minutes/hours and would not be used again, or on a bimodal mode where the nuclear reactor would also be used to provide electrical energy into the system. This last option allows for the hydrogen to be kept cryogenically cooled for long period of time and therefore be reused at a later phase of the mission. Both systems were envisaged and will be described here.

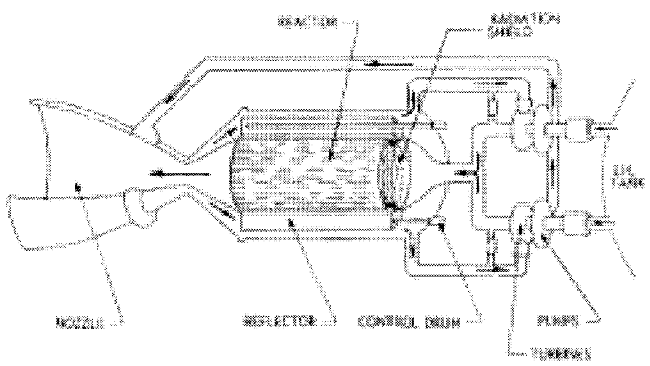

Figure 7: Solid Core NTR Concept (Dual Turbopumps, Expander Cycle).

Trajectory
All the NTP trajectories were run by Leonard Dudzinski at NASA GRC.

Regular NTP

The launch vehicle delivers the spacecraft to 2500-km altitude LEO. The NTP module then injects the spacecraft into an escape trajectory to a C3 of 130 km^2/s^2. The ΔV provided by this burn is 8.43 km/s including 330 m/s of gravity losses. After the vehicle is on its way, the NTP module is dropped and the Orbiter/Lander continues its flight to Titan where it will perform aerocapture. The flight time is 6.4 years.

The arrival Vinf at Saturn is 6.0 km/s and at Titan 4.8 km/s. The entry velocity (inertial) at Titan is 5.4 km/s at 800 km altitude. Here again, the aerocapture system mass fraction was 20% of the total entry mass.

The launch vehicle is the Delta 4450 with a launch capability of 8550 kg at 2500-km LEO.

Bimodal NTP

In this case, the trajectory starts from a C3 of 0 km^2/s^2. A system trade was performed during the study and this option was shown to provide more payload capability than starting from a 2500-km LEO orbit. Here again the NTP system injects the spacecraft to a C3 of 130 km^2/s^2 to reach Saturn 6.4 years after. The injection ΔV is 5.3 km/s (includes 400 m/s for gravity losses). Upon arrival, the NTP system is turned on again. The ΔV for Titan's orbit insertion is about 4.9 km/s including gravity losses.

The launch vehicle is the Delta IV Heavy with a launch capability of 8380 kg at C3 = 0 km^2/s^2.

Systems
The nuclear power source was mounted on one end of the spacecraft. The Orbiter/Lander (all avionics) were located at the opposite end to minimize the radiation dose (see Figure 8). The separation is about 12 m.

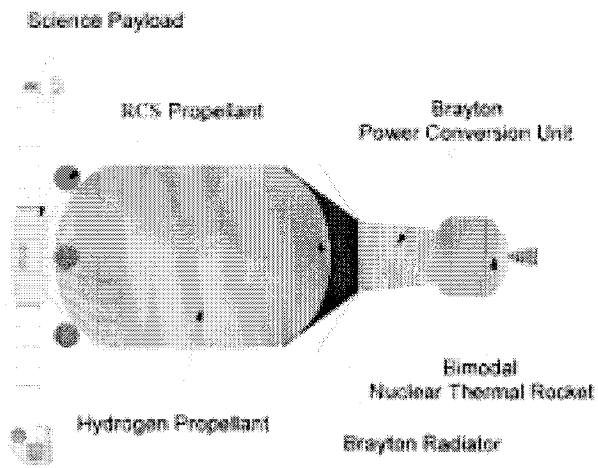

Figure 8: NTP Vehicle Configuration

The NTR engine was gimbaled with 2 DOF relative to the vehicle. The gimbals were actively controlled for pitch and yaw, and hydrazine thrusters were used for roll control. Due to the relatively large size of the vehicle, a scan platform was added to provide the necessary pointing accuracy required by the science instrument.

A Stirling system was used to generate electric power. It provided about 2 kW electric primarily for refrigeration of the Hydrogen, and for spacecraft housekeeping.

In addition, after the reactor has been used for the burn, it continues to produce power for some time. In order to run out the generated radioisotopes during that phase down, an extra 2% propellant was carried.

Tables 12 and 13 show a summary of the spacecraft mass breakdown.

Table 12: NTP Propulsion Module and Spacecraft Mass Breakdown – Regular Case

Subsystem	Mass (kg)	Comment
Payload	1110	
ACS/Avionics	5	
Power System	20	
NTR engine, Feed S.	134	1 engine
Tanks	301	TF = 5.6%
Total Structure	673	
Cabling	52	
Thermal	132	
Aeroshell System	280	
Total:	**1597**	
Contingency (30%)	395	
Propellant	5389	5% conting.
Launch Total:	**8491**	LV = 8550 kg

Table 11: NTP Propulsion Module and Spacecraft Mass Breakdown – Bimodal Case

Subsystem	Mass (kg)	Comment
Payload	850	
ACS/Avionics	5	
Power System	265	
Extractn & Shield	183	
Power conversion	50	
Heat rejection	10	
Power managemt	22	
NTR engine, Feed S.	134	1 engine
Tanks	296	TF = 5.6%
Total Structure	655	
Cabling	54	
Thermal	260	
Total:	**1669**	
Contingency (30%)	501	
Propellant	5279	5+2% conting.
Launch Total:	**8299**	LV = 8380 kg

7. Aerocapture

Aerocapture is an atmospheric flight maneuver executed upon arrival at another planet in which atmospheric drag is used to decelerate the spacecraft into orbit. Aerocapture puts the spacecraft almost immediately into its working orbit. The spacecraft (Orbiter and Lander) is enveloped into a large conical thermal protection system (TPS). No investigations were made to define the type of material and guidance system needed to perform aerocapture. The only figure of merit that was used to define this technology was the mass of the system taken as a percentage of the total entry mass. More details on aerocapture technology are given in [10].

In this study it was found that aerocapture made a significant difference (smaller launch vehicle, larger payload mass fraction) for all but the NEP and NTP bimodal cases. Table 14 shows the amplitude of these benefits.

Table 14: Benefits of Aerocapture Combined with Various Propulsion Systems

Option	Trip time (yrs)	Vinf Saturn (km/s)	TPS mass fractn	Payld mass fractn
Chem AP	10.5			10%
Chem AC	8.5	9.0	25%	23%
SEP AP	7			~14%
SEP AC	5	9.1-9.3	25%	29%
Sol. Sail AP	10.4			~13%
Sol. Sail AC	8.4	9.0	25%	29%
M2P2 AP	7.5			~15%
M2P2 AC	5.6	5.9	20%	31%

AP: All Propulsive, AC: Aerocapture

Propulsion Technologies Trade Results

The goal of this study was to assess the benefits of a limited set of advanced technologies for a Titan Explorer Mission and mainly for the three following figures of merit: trip time, launch vehicle and payload mass fraction. Other figures of merit such as technology development cost, mission cost, operational complexity, sensitivity to malfunctions, reliability and safety, development time... were assessed on a broader scale during the IISTP task and will not be discussed here. Table 15 summarizes the mission results for each of the propulsion options. Here the payload mass fraction is defined as the dry mass of the spacecraft not including the propulsion module (nor aerocapture system), which was 850 kg for most cases, divided by the total launch mass.

As a first general comment, since the goal for each propulsion system option was to use the smallest possible launch vehicle with the shortest possible trip time, there was no effort to look at other ways of using the proposed technologies to reduce their cost to the mission project. For instance, SEP solutions exist that would significantly reduce the SEP power needed and therefore solar array cost but increasing the trip time. Once again, the objectives of this study were not to optimize a propulsion system for a mission, but to build a common ground against which the benefits of these technologies could be compared.

As a second general comment, since the Orbiter and Lander design and science payload stayed the same for each propulsion option, this study did not capture the other mission benefits that a nuclear system could provide. In this Orbiter/Lander design, there was no use of the vast amount of power that a nuclear system could offer once at Titan.

Table 15: Numerical Comparison of the technologies studied.

	Chem AP	Chem AC	SEP 5-kW AC	SEP 10-kW AC	Solar Sail AC	M2P2 AC	NEP	NTP Bi-modal	NTP AC
Departure	C3=11	C3=11	C3=10	C3=6	C3=0	C3=0	LEO		
Trip time (yrs)	10.5	8.4	5.2	5.5	8.5	5.6	7		
Launch Vehicle	Titan IV	A 531	D4240	D4240	D4240	D4240	D4450		
Payload mass (kg)	850	850	850	850	878	850	850		
Launch mass (kg)	8272	3675	2912	3246	3041	2732	~8000		
Propellant mass (kg)	6126	1667	527	804	20	462	~3000		
Payload mass fraction	10%	23%	29%	26%	29%	31%	~11%		

AP: All Propulsive, AC: Aerocapture

Conclusions

This paper describes the technology assumptions and mission results of the evaluation of 6 advanced propulsion technologies for the Titan Explorer Mission. The 6 technologies evaluated were: Solar Electric Propulsion (SEP), Solar Sails, Mini-Magnetospheric Plasma Propulsion (M2P2), Nuclear Electric Propulsion (NEP), Nuclear Thermal Propulsion (NTP), and Aerocapture.

This study shows that Aerocapture greatly enhance 5 out of 7 of the mission architectures by increasing their payload mass fraction from an average of about 13% to an average of about 29% and, in the case of chemical propulsion, by enabling the use of a smaller launch vehicle.

The Solar Electric Propulsion options provide the optimum combination of a short trip time and high payload mass fraction.

The M2P2 technology looks very promising. A clear benefit to this technology is that it is potentially a very simple system. However, it is still at a very low TRL level and there are still many unknowns with respect to the basic physics, the approach to the trajectory control, and to the validity of the assumptions that were used in this study. A better understanding of all those issues should help us within the next few years refine our mission benefit evaluations.

And finally, the Titan Explorer mission was probably not demanding enough in terms of ΔV or energy requirements, to show the full benefits of NEP, NTP, or sails. Other missions evaluated during the IISTP task might reflect their niche of applicability in a better way.

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References

[1] Roy Kakuda, JPL, Personal Communication.

[2] Advanced Projects Design Team, JPL, IISTP 03-01 Final Report.

[3] T. Sweetser, R. Maddock, J. Johannesen, J. Bell, P. Penzo, A. Wolf, S. Williams, S. Matousek, S. Weinstein, "Trajectory design for a Europa Orbiter Mission: A Plethora of Astrodynamic Challenges," Paper AAS 97-174, AAS/AIAA Space Flight Mechanics Meeting, Huntsville, Alabama, February 1997.

[4] J. E Polk, et al., "Validation of the NSTAR Ion Propulsion System on the Deep Space One Mission: Overview and Initial Results," AIAA 99-2274, presented at the 35th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Los Angeles, CA, June 1999.

[5] C. Garner, H. Price, D. Edwards, R. Baggett, "Development and Activities in Solar Sail Propulsion", AIAA 2001-3234.

[6] M. Leipold, et al., "Solar Sail Technology Development and Demonstration", 4th IAA International Conference on Low Cost Planetary Missions, Laurel, Maryland, May 2-5 2001.

[7] R. M Winglee, J. Slough, T. Ziemba, and A. Goodson, "Mini-Magnetospheric Plasma Propulsion: High Speed Propulsion Sailing the Solar Wind," Space Technology and Applications International Forum-2000, edited by M. S. El-Genk, American Institute of Physics CP504, 1-56396-9, p. 962, 2000.

[8] M. Noca,, J. Polk, R Lenard, "Evolutionary Strategy for the use of Nuclear Electric Propulsion in Planetary Exploration", 18th Symposium on Space Nuclear Power and Propulsion, STAIF Conference, February 2001, Albuquerque, New Mexico.

[9] R. J. Lipinski, S. A. Wright, M. P. Sherman, R. X. Lenard, R. A. Talandis, D. I. Poston, R. Kapernick, R. Guffey, R. Reid, J. Elson, and J. Lee, "Small Fission Power Systems for NEP", proposed at the 19th Symposium on Space Nuclear Power and Propulsion, STAIF Conference, February 2002, Albuquerque, New Mexico.

[10] Hall, J. L., "A Review of Ballute Technology For Planerary Aerocapture," Presented at the 4th IAA Conference on Low Cost Planetary Missions, Laurel, MD, May 2-5, 2000.